Advanced life assessment of the lead crack configuration of the RNLAF F-16 wing damage enhancement test

Customer

Defence Materiel Organisation

NLR-TP-2015-140 - April 2015

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UNCLASSIFIED

EXECUTIVE SUMMARY

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Problem area

The actual usage of the RNLAF (Royal Netherlands Air Force) and the Fuerza Aérea de Chile (FACh) F-16 Block 15 differs from the original design spectrum. The more severe usage will reduce the service life. To extend the (economic) life, the F-16 was subjected to several modification programs. However, structural modifications mainly involved the fuselage frames and modifications with respect to the wing were limited. Based on analyses and in-service experience Lockheed Martin (LM) defined damage prone locations and performed damage tolerance and durability analyses for these selected areas. However, damage locations not accounted for and/or damage propagation exceeding the LM predictions, may hinder a safe and economic operation of the fleet until their planned retirement dates. To determine whether structural modifications or new wings are required to ensure safe and economic operation, it is necessary

Report no.
NLR-TP-2015-140

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Report classification
UNCLASSIFIED

Date
April 2015

Knowledge area(s)
Health Monitoring & Maintenance of Aircraft

Descriptor(s)
Life (durability)
Damage Tolerance
Stress Intensity Factor
Fatigue Testing
Finite Element Method (fem)

This report is based on a presentation to be held at the International Committee on Aeronautical Fatigue and Structural Integrity (ICAF), Helsinki, June 3-5, 2015.

to determine the current damage condition and the expected damage development.

Description of work

In order to verify the current estimates for the service life and maintenance requirements of the RNLAF and FACh F-16 Block 15 wings, a durability test has been conducted at NLR on the left hand side wing of a decommissioned F-16 Block 15 aircraft of the RNLAF that had accumulated 4,200 operational flight hours. This durability test aimed to grow in-service cracks of sub-detectable size to a size where they could readily be detected. The main objective of the test was to determine if the ex-service wing contained damage not accounted for in the durability test programme of the 1970s by General Dynamics or in the analyses of Lockheed Martin. Other objectives were to generate data that can be used for an assessment of the current maintenance programme and to establish the most critical locations and the most likely fail scenario.

A lifing framework was developed at NLR to enhance the life prediction for complex geometries and/or complex loads by improving the accuracy of the applied stress intensity factor solution, the computation of which can be very time consuming. Due to the increase in computer capacity, the better integration of computer aided design and finite element tools, and new numerical algorithms, the computation of a stress intensity factor solution today can be done much more efficiently. A framework was developed with which a crack growth life analysis for a complex geometry and load can be performed much more efficiently, automating the different steps as much as possible, especially the time consuming parts. The framework provides a coupling of an (integrated) CAD tool to define parameterised crack geometries, a finite element tool to compute the stress intensity factor solution and an advanced crack growth tool to compute the life of the structure.

Results and conclusions

Upon completion of the test the wing had been subjected to a load spectrum that was equivalent to a total of 18,200 RNLAF flight hours. During the test a number of fatigue cracks had initiated and grown, some of them to a significant size. The wing was able to sustain the three limit load cases that were applied after the durability test, however.

The lifing framework was successfully applied to the complex lead crack configuration that occurred in the full scale fatigue test on an F-16 Block 15 wing, yielding an accurate damage tolerance life estimate.

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The contents of this report may be cited on condition that full credit is given to NLR and the authors.

This publication has been refereed by the Advisory Committee AEROSPACE VEHICLES.

Customer	Defence Materiel Organisation
Contract number	DMO-15-120-11-7011-01
Owner	National Aerospace Laboratory NLR
Division NLR	Aerospace Vehicles
Distribution	Unlimited
Classification of title	Unclassified
Date	April 2015

Approved by:

Author	Reviewer	Managing department
F.P. Grooteman	M.J. Bos	H.G.S.J. Thuis
Date 2-4-2015	Date 2/4/2015	Date 7-4-2015

Summary

In order to verify the current estimates for the service life and maintenance requirements of the RNLAF and FACh F-16 Block 15 wings, a durability test has been conducted at NLR on the left hand side wing of a decommissioned F-16 Block 15 aircraft of the RNLAF that had accumulated 4,200 operational flight hours. This durability test aimed to grow in-service cracks of sub-detectable size to a size where they could readily be detected. The main objective of the test was to determine if the ex-service wing contained damage not accounted for in the durability test programme of the 1970s by General Dynamics or in the analyses of Lockheed Martin. Other objectives were to generate data that can be used for an assessment of the current maintenance programme and to establish the most critical locations and the most likely fail scenario. Upon completion of the test the wing had been subjected to a load spectrum that was equivalent to a total of 18,200 RNLAF flight hours. During the test a number of fatigue cracks had initiated and grown, some of them to a significant size. The wing was able to sustain the three limit load cases that were applied after the durability test, however.

A lifing framework was developed at NLR to enhance the life prediction for complex geometries and/or complex loads by improving the accuracy of the applied stress intensity factor solution, the computation of which can be very time consuming. Due to the increase in computer capacity, the better integration of computer aided design and finite element tools, and new numerical algorithms, the computation of a stress intensity factor solution today can be done much more efficiently. A framework was developed with which a crack growth life analysis for a complex geometry and load can be performed much more efficiently, automating the different steps as much as possible, especially the time consuming parts. The framework provides a coupling of an (integrated) CAD tool to define parameterised crack geometries, a finite element tool to compute the stress intensity factor solution and an advanced crack growth tool to compute the life of the structure. The framework was successfully applied to the complex lead crack configuration that occurred in the full scale fatigue test on an F-16 Block 15 wing of the RNLAF, yielding an accurate damage tolerance life estimate.

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Content

Abbreviations

Acronym	Description
ASIP	Aircraft Structural Integrity Program
BEM	Boundary Element Method
BL	Butt-Line
CAD	Computer Aided Design
cgFEM	coarse grid Finite Element Model
DADTA	Durability And Damage Tolerance Analysis
DMO	Defence Materiel Organisation
DVI	Detailed Visual Inspection
FACh	Chilean Air Force
FEM	Finite Element Method
FH	Flight Hour
FSMP	Fleet Structural Maintenance Plan
LEF	Leading Edge Flap
LL	Limit Load
LM	Lockheed Martin
NDI	Non-Destructive Inspection
RNLAF	Royal Netherlands Air Force
QF	Quantitative Fractography
SIF	Stress Intensity Factor
XFEM	eXtended Finite Element Method

1 Introduction

The Defence Materiel Organisation (DMO) of the Netherlands has tasked the National Aerospace Laboratory NLR to conduct a so-called damage enhancement test on the LHS wing of a decommissioned F-16 Block 15 of the Royal Netherlands Air Force (RNLAF). The wing had a known history and had accumulated 4,200 in-service flight hours. The results of the test programme are relevant for both the RNLAF and the Chilean Air Force (FACh), who shared the programme results with the RNLAF on a government to government basis.

The damage enhancement test essentially was a durability test (and will be referred to as such in the remainder of the paper) that aimed to grow in-service cracks of sub-detectable size to a size where they may readily be detected. This significantly enhanced the teardown inspection programme that was conducted in parallel on the RHS wing of the same aircraft. The main objective of the test was to determine if the tested ex-service wing contained damage not accounted for in the early durability test programme that was performed in the late 1970s by General Dynamics or in the current durability and damage tolerance analysis (DADTA) of Lockheed Martin (LM). Other objectives were to generate data (i.e. crack growth curves for the locations that are most critical) that can be used for an assessment of the current aircraft structural integrity program (ASIP) inspection programme and to establish the most likely fail scenario, incl. an estimate of the associated technical end of life time.

A damage tolerance analysis was performed for the lead crack configuration of the damage enhancement test. The result of such a damage tolerance life prediction is in particular sensitive for the applied loads and stress intensity factor (SIF) solution. Loads are measured extensively nowadays often even on individual aircraft, as in the case of the RNLAF. In the DADTA use is made of handbook SIF solutions. An overestimation of the SIF by 25% can yield a factor 2 reduction in life, yielding an extensive reduction in operational life. Hence, most significant improvements can be obtained in the applied stress intensity factor solution. For the F-16 lead crack growth analysis therefore accurate SIF solutions were applied. For this an advanced lifing framework was applied that has been developed at NLR to enhance the crack growth life prediction for complex structures and complex loads. The main objective was to automate the different steps as much as possible, especially the time consuming parts such as the stress intensity factor (SIF) solution computation.

2 F-16 wing damage enhancement test

The durability test covered 18,200 flight hours (incl. the 4,200 in-service hours of the tested wing), which is somewhat more than the minimum of two nominal lifetimes of 2 x 8,000 = 16,000 flight hours that is specified by MIL-STD-1530C [1]. In order not to miss any potentially critical cracks because of insufficient loading conditions, the test setup was fairly complex and involved the use of 23 hydraulic load actuators and the application of a representative fatigue load spectrum that was based on the 2004 L/ESS spectrum as used by LM in the development of the current fleet structural maintenance plan (FSMP) for the RNLAF. The added advantage was that the observed crack growth rates in the tested wing were representative of those in service. This enabled the translation of the test results to service conditions and provided an experimental validation of the theoretical crack growth curves provided by LM in the DADTA. It is emphasized that the results are not meant for certification purposes, however. The available budget and time precluded the presence of the leading edge flap (LEF) and flaperon (the calculated interface loads have been applied, however) and the pressurization of the fuel tank. The durability test involved a simplification by considering the wing only configuration; the design of a representative wing root support has been dealt with using calculations with the F-16 Block 15 coarse grid Finite Element Model (cgFEM) of LM that is available at the NLR.
The test has been carried out under room temperature ambient conditions. No artificial damages have been applied. Only naturally existing damages, caused by in-service fatigue loading, corrosion, etch pits, tool marks, etc. were considered.
No formal residual strength test has been conducted at the conclusion of the test; three different limit load cases have been applied, however.

Prior to the installation of the test article in the test rig, all wing-specific inspections as called out in the Fleet Structural Maintenance Plan have been performed where possible.

Figure 1 provides an impression of the test setup. The static and fatigue loads on the wing box were introduced by a total of 23 force controlled hydraulic actuators. With these actuators, forces were applied that were representative of the aerodynamic loads, the inertia loads (incl. those from the stores), the LEF loads at the four rotary actuators and the brake, and the flaperon loads at the three flaperon hinges. The simulated aerodynamic loads were introduced at relevant wing stations with two hydraulic actuators per station, to enable simultaneous introduction of wing bending loads and torsion (also called 'pitch') loads. The general layout and the numbering are indicated in Figure 2.

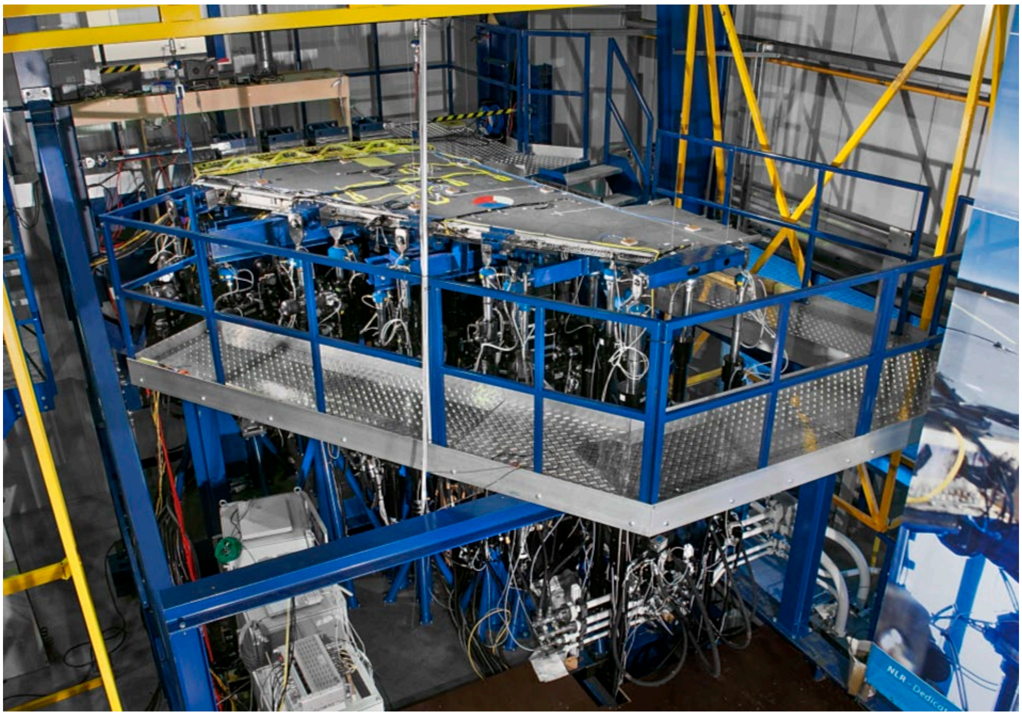

Figure 1: Impression of the F-16 Block 15 wing durability test setup. The yellow frame is the measurement frame to which the six displacement transducers on the wing upper surface were connected.

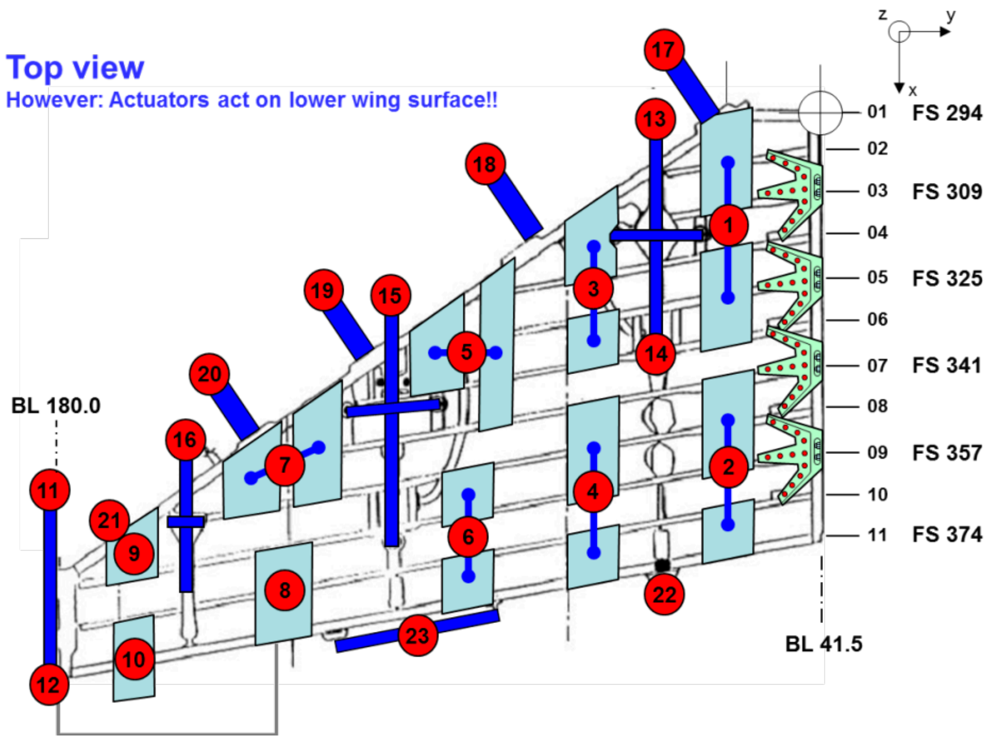

Figure 2: Numbering and position of the hydraulic actuators; the coordinates are in the F-16 reference frame. The red dots indicate the actuator positions, the grey areas represent the load pads and the blue lines represent the dummy store pylons, dummy LEFRAs and whiffletrees.

The following groups of actuators were distinguished:

- Actuators 1 to/incl. 10 were used to introduce the simulated aerodynamic loads and wing inertia loads on the lower surface of the wing. They were connected to the wing by means of bonded load pads, which consisted of rubber pads that were sandwiched between the wing skin and steel brackets. The brackets were interconnected by means of whiffletrees, to ensure a uniform distribution of the actuator loads.
- Actuators 11 to/incl. 16 were used to introduce the simulated aerodynamic and inertia loads from the stores and launchers under the wing. They were connected to the wing by means of steel dummy brackets. The bracket-wing interfaces were representative of the real launcher hardware.
- Actuators 17 to/incl. 21 were used to introduce the simulated aerodynamic and induced bending loads from the LEF. Actuators 17 to/incl. 20 were connected to the wing by means of steel dummy brackets with a wing interface that, in terms of stiffness, was representative of the rotary actuators. Actuator 21 was directly connected to the wing part of the LEF hinge bracket.
- Actuators 22 and 23 were used to introduce the simulated aerodynamic and induced bending loads from the flaperon, using a sliding pin and a dummy bracket respectively.

The actuator loads were measured with calibrated load cells that were placed between the actuators and the wing. The load trains (i.e. all elements between the wing brackets and the test frame, including the actuators and load cells) included spiral washers to ensure proper fixation.

The test article was installed in a closed self-contained steel test frame which had a steel floor on which the hydraulic actuators were mounted. The test article was connected to four steel H-beams that were vertically mounted in the test frame in such a way that they were statically determinate. These Wing Attachment Beams (WABs) simulated the center fuselage carry-through bulkheads of the F-16. It has been verified by means of FEA that this setup was such that the distribution of the wing interface loads over the simulated bulkheads was sufficiently similar to that in the real aircraft. The upper side of each of the vertical beams was connected to the test frame by means of an adjustable link. Each link was instrumented with a calibrated strain gauge bridge. The outputs of the four bridges have been monitored during the durability test to verify the proper distribution of the transferred bending moments over the four simulated bulkheads during the test. The lower flanges of the vertical beams were connected to the test frame through instrumented pins in spherical plain bearings that measured the loads in y- and z-direction. These loads were monitored as well. The shear ties at the front and rear spars of the wing were fixed to the test rig by means of vertical steel straps and the upper sides of the forward and aft WABs were fixated to the test rig by means of rod ends. The reaction loads in these straps and rod ends were continuously monitored with load cells.

The instrumentation on the test article consisted of strain gauges and displacement transducers. The strain gauges were placed such that all relevant load paths were covered.

Prior to the start of the durability test, a series of static commissioning load cases have been applied to verify the correct functioning of the test setup. In addition, strain surveys have been performed regularly during the durability test programme to monitor the quality of the test and to detect a possible change of load paths due to the growth of fatigue cracks. The same load cases have been used for the commissioning test and the strain surveys. The load cases were selected from the set of 211 Limit Load (LL) cases that LM has provided with the F-16 Block 15 coarse grid FEM. They were scaled down to 50-80% of the LL values, depending on the load case.

The flight-by-flight load spectrum consisted of a series of load lines or condition codes, where each condition code represented a set of actuator loads that simulate a point in the sky. The spectrum has been developed and validated using the cgFEM of the F-16 Block 15 (which is sufficiently detailed as to resolve the relevant load paths), the associated set of 211 static limit load cases, the NLR developed tool AELEV-MIL (for the calculation of the operational flight loads [2]) and the NLR developed spectrum generation software CLASS [3]. The spectrum represented a block of 500 simulated flight hours and 412 flights that was repeatedly applied up to the end of the durability test. The 500-flight hours spectrum contained 117,732 load lines, which is in line with the spectrum used by LM in the F-16 Block 50 full-scale durability test [4]. The process to generate the load spectrum is outlined in figure 3. The input to the process was formed by the set of measured flight parameters for the 2004 L/ESS batch of 412 flights (500 flight hours) that has been used in the development of the current FSMP for the RNLAF fleet.

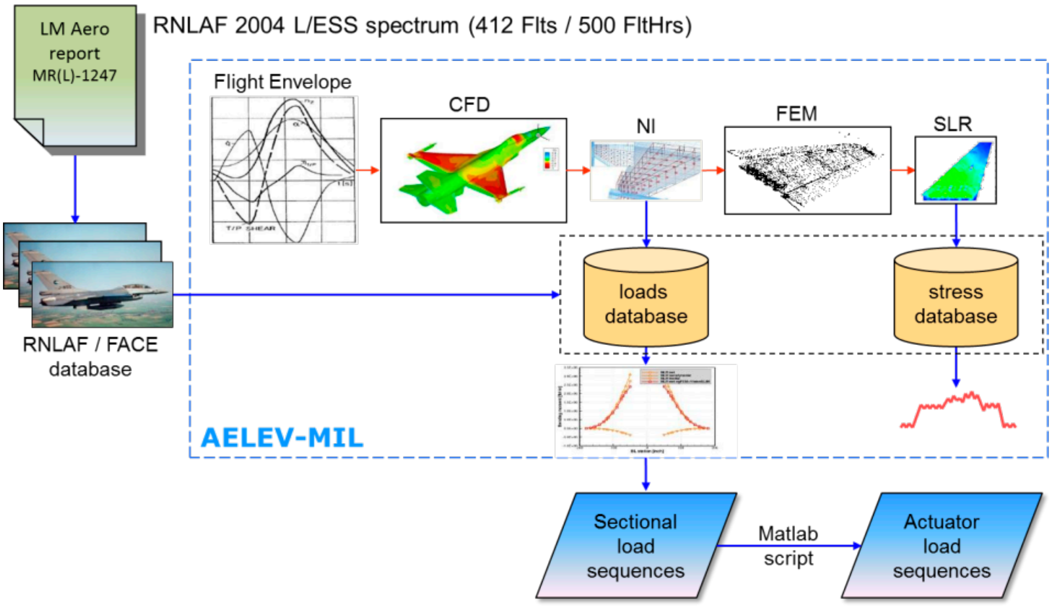

Figure 3: Process to generate the actuator load spectrum used in the test.

The actuator load spectrum included three different so-called marker load blocks to improve the readability of the fatigue crack surfaces; this greatly enhanced the post-test Quantitative Fractography (QF) efforts, see for instance figure 4. It was verified that the addition of the marker loads does not significantly affect the severity of the load spectrum in terms of fatigue life or crack growth life (and that the small influence is conservative).

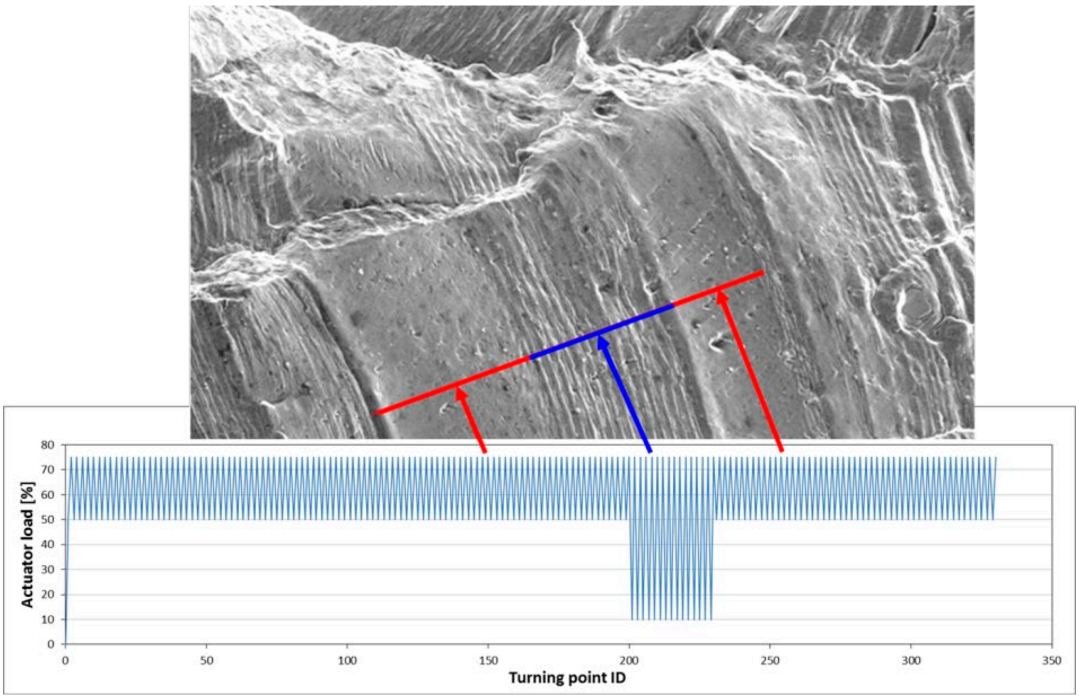

Figure 4: Definition of marker load sequences and corresponding fracture surface pattern.

Figure 5 shows the maximum deflections of the wing tip in upward and downward direction. Measured deflections at actuator #12 (dummy tip launcher aft position) were +435 mm up and -255 mm down, with a total range of 690 mm.

Figure 5: Composite photographs that show the wing deflections during the applied limit load cases "-3g down bending" and "+9g up bending".

During the test a number of fatigue cracks have initiated and grown, some of them to a significant size. The wing was able to sustain the three limit load cases that were applied after the durability test, however.

The lead crack configuration that occurred in the test consisted of a crack in the lower skin starting at the rectangular cutout at Butt-Line (BL) 71, depicted in figure 6, growing to the spar #6 fasteners and finally to the pylon hardpoint hole, and exactly matched the assumed crack configuration in the DADTA for this location, depicted in figure 11. During this period, other relevant cracks started at nearby locations in the rectangular cutout, pylon hardpoint hole and in the underlying spar #6, figure 7, interacting with this crack.

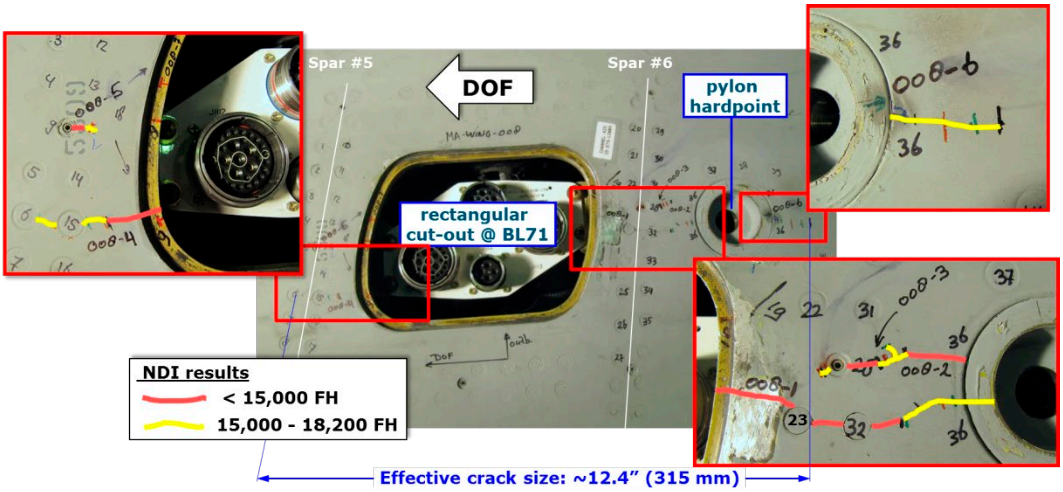

Figure 6: Widespread Fatigue Damage in the area of ASIP CP MA-WING-008: "Lower wing skin, pylon Rectangular Cutout at BL 71, Cutout and Fastener Holes". A total of ten cracks were present after completion of the durability test. The cracks have been highlighted with red (initial phase) and yellow (final phase).

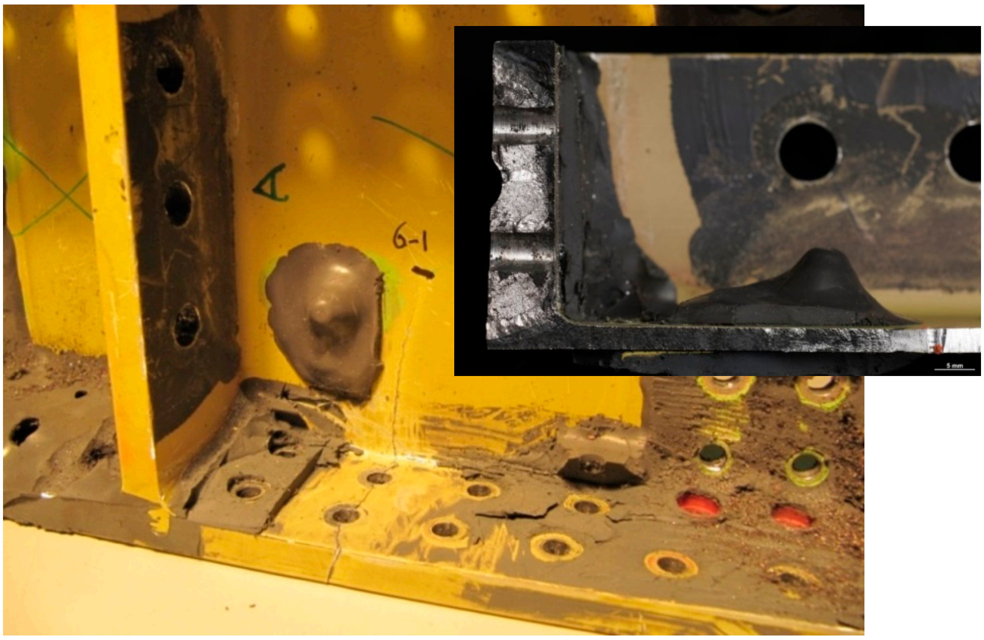

Figure 7: F-16 Block 15 Wing spar #6 crack.

After removal of the test article from the test rig, the upper wing skin was removed at Leeuwarden air force base. Upon return of the wing and upper skin at NLR in Marknesse an extensive detailed visual inspection (DVI) and non-destructive inspection (NDI) was performed and a number of additional cracks were found. The largest of the detected cracks were opened for fractographic examination. Summaries of the most significant findings are provided in figure 8.

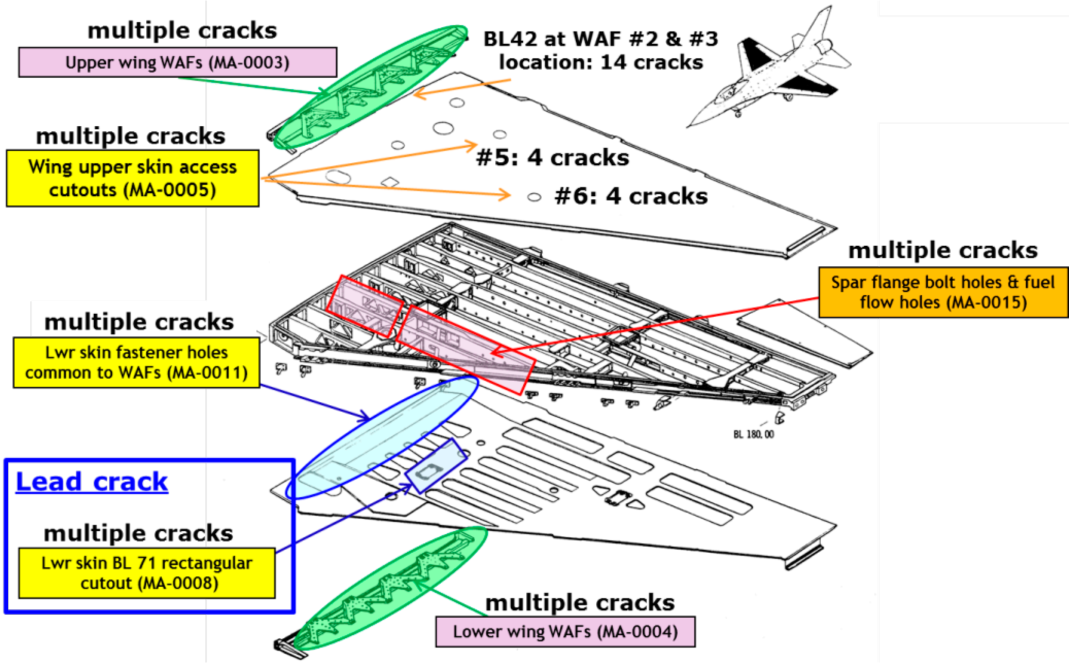

Figure 8: Summary of the most significant fatigue damages detected during inspection after the durability test.

The loads and strains that have been measured during the static tests (i.e. strain surveys and final LL test) and the durability test have been used in a correlation study to verify the following points:

1. the validity of the available F-16 Block 15 cgFEM;
2. the consistency and quality of the applied loads in the durability test;
3. the validity and relevance of the durability test itself, by comparison of the local loading conditions at fatigue critical locations in the wing as installed in the test rig to those in the wing as installed on a real F-16 Block 15 aircraft.

The results confirmed each of the above three points.

3 Lifing framework

Fatigue is one of the main design drivers for aircraft structures and airworthiness regulations require proof that aircraft can be operated safely. This implies that critical components must be replaced or repaired before safety is compromised. The philosophies underlying the approaches for guaranteeing safety are called Safe-Life and Damage Tolerance. The Damage Tolerance philosophy recognises that damage can occur and develop during the service life of a component. Also, it assumes that cracks or flaws can be present in pristine structures. Safety is obtained from this approach by the requirements that either (1) any damage be detected by routine inspection before it results in a dangerous reduction of the static strength (inspectable components), or (2) initial damage shall not grow to a dangerous size during the service life (non-inspectable components). For Damage Tolerance analysis to be successful it must be possible to predict crack growth during the time until the next inspection or until the design service life is reached.

The DADTA of Lockheed Martin for the F-16 is based on a crack growth model in which the life in cycles or flight hours is computed to grow a crack from an initial flaw up to a critical crack size causing fracture of the structural part or functional impairment (e.g. fuel leakage). The crack growth model uses the stress field in front of the crack tip. It can be shown that this stress field is of the following form, in polar coordinates, see for example [5]:

$$\sigma(r,\theta) = \frac{K_I}{\sqrt{2\pi r}} f^I(\theta) + \frac{K_{II}}{\sqrt{2\pi r}} f^{II}(\theta) + \frac{K_{III}}{\sqrt{2\pi r}} f^{III}(\theta) + HOT \tag{1}$$

where r is the radius denoting the distance from the crack tip and θ the angle with respect to the crack surface. Three different crack opening modes (tension, in-plane shear and transverse shear) can be distinguished with corresponding stress intensity factors (SIF) K. The tensile mode-I is by far the most important. SIFs for the same mode may be added (superposition), but not for different modes. For homogeneous, isotropic materials the following relation exists for combined mixed-mode SIF:

$$G = (1 - \nu^2)\left(\frac{K_I^2 + K_{II}^2}{E}\right) + (1 + v)\frac{K_{III}^2}{E} \tag{2}$$

in which G is the energy release rate and represents the energy per unit new crack area due to an infinitesimal crack extension. The mode I SIF solution is the most important since the crack in general will quickly change direction perpendicular to the load direction, becoming a mode I case only.

The SIF solution forms the basis of all linear elastic crack growth analyses. More in general, the SIF can be written as:

$$K_I = \beta(a) S\sqrt{\pi a} \quad (3)$$

where the function $\beta(a)$ is a factor correcting for the difference in geometry with respect to the infinite plate and is called the normalised SIF, i.e. $\beta(a) = K_I/K_0$. K_0 is a normalisation SIF which for a remote stress is often selected equal to the infinite plate solution $K_0 = S\sqrt{\pi a}$.
For simple geometries (mostly flat plates with straight cracks) and loading conditions (mostly uniform stress, bending stress or point loads) many (semi)-analytical solutions for mode-I have been generated and collected in handbooks such as [6-8].
Besides simple geometries and loading conditions, most handbook solutions are for mode-I solutions under load control. There exist only a very limited number of mode II and mode III solutions and also only a few so-called displacement controlled solutions, were the applied load is a remote displacement instead of a remote stress.
To apply these handbook solutions to real structures, it has to be simplified in geometry and loading. By means of combining different solutions (superposition), somewhat more complicated geometries and loading conditions can be approximated. A well-known approach is the compounding method [9].

The crack growth life is determined by computing the crack growth during a load cycle from the minimum to the maximum load, causing a change in the SIF, i.e. $\Delta K = K_{max}$-K_{min}. The most simple, but often applied, crack growth equation is the Paris equation:

$$\frac{da}{dN} = C\Delta K^m \quad (4)$$

Other more advanced equations exist, such as the Forman and NASGRO equation. The parameter m in general has a value around 3.0, indicating that *an overestimation of the SIF by 25% can result in a factor 2 reduction in life*, yielding an extensive reduction in operational life. Equation (4) indicates that such an error can be easily introduced by the use of an erroneous normalised SIF solution β and/or by an error in loads. Loads are measured extensively nowadays often even on individual aircraft like in the case of the RNLAF F-16s. Hence, most significant improvements in the life prediction can be obtained in SIF values.

A National Technology Project has been performed with the objective to enhance the life prediction for complex geometries and/or complex loads by improving the accuracy of the applied normalised SIF solution**.** This can be done with the aid of numerical methods like the finite element method (FEM) and the boundary element method (BEM). In this way, more accurate SIF solutions can be obtained for complex geometries and loads. FEM is more versatile (BEM for instance is less suitable for anisotropic materials and non-linear elastic-plastic problems) and widely available and was therefore selected as the preferred numerical method. The calculation of a SIF solution can be very time consuming. Due to the increase in computer capacity, the better integration of computer aided design (CAD)/FEM and new numerical algorithms, the computation of a SIF solution today can be done much more efficient. The calculation of the crack growth life using a SIF solution computed with FEM requires the following steps:

1. Generation of the baseline finite element model, often already available
2. Compute the SIF solution for several crack sizes
    a. Insert the new crack length(s)
    b. Regenerate the mesh, e.g. focused mesh at crack tip(s)
    c. Compute the stress/strain distribution
    d. Post-processing of the results to derive the SIF
3. Importing the SIF solution(s) in a crack growth tool
4. Perform the life analysis

The objective was to automate each of these steps as much as possible, especially the time consuming parts, and couple the required tools. This resulted in a framework with which a life analysis can be performed much more efficiently.

To obtain a much more automatic generation of SIFs the FEM model has to be parameterised, i.e. the crack geometry (and even better also the part geometry) has to be defined in terms of modifiable parameters, allowing automatic changes to the crack geometry, i.e. to simulate crack growth. The right way to do so is by means of coupling the FEM tool with a CAD tool. The combination CATIA and Abaqus was examined for this purpose, both from the same software vendor. A so-called associative interface is offered with which modifications made to the CAD model are automatically transferred to the Abaqus finite element model. However, this only has very limited functionality and is by far not (yet) a viable option for the following main reasons:

- Materials are imported but are not assigned to parts, which has to be done manually
- Only parts are imported but no assembly, which still has to be created in Abaqus
- Boundary conditions and loads are not imported
- There exist no equivalent type for a partition used to model the crack or facilitate the generation of the finite element mesh

- Cracks cannot be modelled in CATIA and transferred to Abaqus
- Modifications made in the CATIA model are not always correctly exported to the Abaqus model and might cause unexpected errors

Nevertheless, the CATIA CAD model can still be used to import an existing geometry of the part(s), serving as a good basis for the creation of the baseline/reference model. Abaqus comes with an internal CAD engine as well that is tightly linked with the finite element model and as such much more suitable. Although limited, it still offers a versatile toolset to construct complicated models. Hence, the integrated CAD engine was applied instead of a CATIA CAD model.

The crack growth life of a structural detail is determined with a crack growth program. Various crack growth programs exist of which the most well-known is NASGRO [10]. A more recent NLR in-house developed crack growth program is Cragro++ [11], written in the object oriented programming language C++, which provides for much better data management, encapsulation, extendibility and maintainability. Existing crack growth tools can perform crack growth analyses for various crack growth laws and models on various simple crack geometries. The available geometries are the most important handbook solutions. To enable more accurate life prediction for complex geometries and/or loads, the SIF solution needs to be determined by FEM (or BEM) and imported in the crack growth tool. The latter can be done in two ways:

1. Calculate the normalised SIF $\beta(a)$ for various increasing crack sizes and import this table of SIF values in the crack growth tool. For intermediate crack sizes the normalised SIF solution is determined by means of interpolation.
2. For more complicated problems a complete table cannot be generated in advance and a SIF solution has to be generated on-the-fly for the crack size(s) at hand. This, for instance, is the case when the crack path is unknown in advance. Another example is crack bifurcation, where a single crack is split in two or more cracks growing in separate directions, which for example occurs in integrally machined stiffened parts, e.g. Chinook frames. A third example is a two- or even three-dimensional crack, such as a corner or surface crack, where the crack front changes during growth, e.g. a crack in a turbine blade. For this, it is necessary to couple the crack growth tool with a CAD/FEM tool. The crack growth program then becomes the driver and requests the CAD/FEM tool to provide a SIF solution for specific crack geometries.

For the above mentioned reasons, a framework, schematised in figure 9, was built to enable a more accurate highly automated life prediction for complex geometries and/or loads. It consists of the crack growth program Cragro++ coupled with the finite element program Abaqus.
Cragro++ was selected for two important reasons. First, it could be extended with an FEM interface that passes the current crack geometry and reads the resulting SIF solution, for which access to the source code is required. Second, it has no limitations in the number of cracks and load cases, important for realistic cases like the lead crack of the F-16 wing test. Third, the specific Lockheed Martin crack growth model is implemented for life analyses for the F-16, other models can be easily added.
Abaqus was selected, because it comes with an integrated CAD environment that enables parameterised models, which is a prerequisite for automatic SIF computation. Second, it has extensive capabilities for SIF computation with both conventional crack tip elements and the more recently implemented extended finite element method (XFEM) approach. Third, it has an extensive application programming interface that provides easy access to the FE model for modifications (e.g. new crack size) as well as the result database to extract and process the (SIF) results.
The conventional crack tip element approach gives the most accurate SIF solution. The XFEM approach is not as mesh independent as sometimes suggested, but still requires a sufficiently fine structured mesh around the crack tip that correctly captures the stress field. Nevertheless, for complex geometries and/or crack shapes XFEM remains a very interesting approach, especially when a number of issues with the applied Abaqus 6.12 XFEM implementation are solved in a future version.

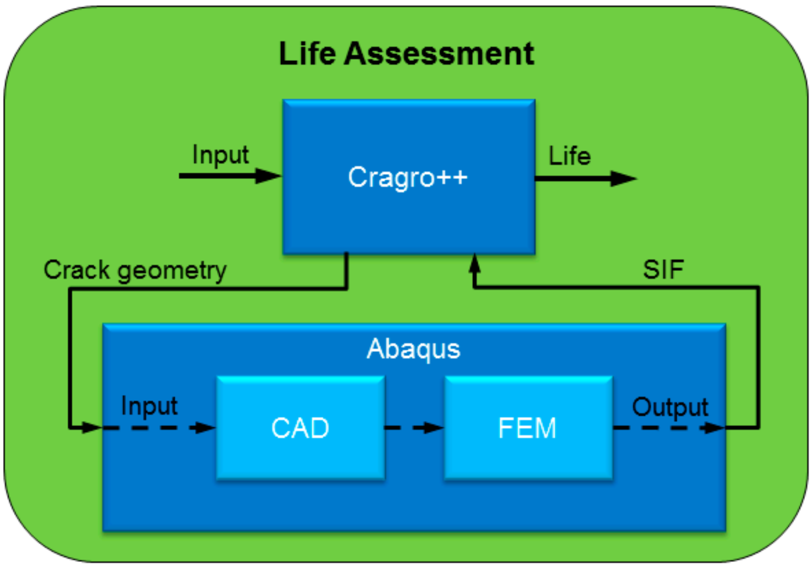

Figure 9: Life assessment framework for complex geometries and loads.

To facilitate an interface with a CAD/FEM tool enabling automatic computation of the SIF solution for a given crack size, a special crack geometry model was implemented in Cragro++. The objective was to provide a very generic interface applicable to as broad a range of problems as possible. Hence, it allows the definition of an unlimited number of cracks, for all three crack opening modes, as well as the definition of constants and variables (expressions). The constants and variables, for instance, allow the definition of any normalisation stress intensity factor K_0, but also the definition of new variables that can be exported to the CAD/FEM tool (here Abaqus), for instance to change mesh properties or dimensions. The latter, for example, can be applied to alter the focused mesh dimensions or to automatically generate a set of SIF solutions for varying geometrical details such as for instance different length over width ratios.

To verify the functioning of the framework it was applied to a variety of available one- and two-dimensional handbook solutions that include different geometries and loads. An example is presented next.

3.1 Central slant crack in a rectangular sheet loaded by a parabolic tensile stress

Here a crack in a rectangular sheet is under an angle with respect to the load, being a parabolic tensile stress distribution, as depicted in figure 10. Since the crack is under an angle it yields a combined mode I and mode II stress intensity factor solution. The handbook solution [7] is depicted in the figure as black dots. The crack size is kept constant and the effect of the height to width ratio is varied, demonstrating the automatic change in geometrical properties besides the crack geometry, requiring a parameterised geometry as well.

The computed normalised SIF solution using 2D and 3D crack tip elements and XFEM are depicted. The 2D crack tip element solution coincides with the handbook solution that assumes a plane stress condition. The 3D crack tip solution is somewhat higher for the mode I SIF, but the mode II solution shows a larger deviation caused by the more realistic non plane stress condition. The mode I XFEM solution is somewhat higher than the crack tip element solution, but the mode II solution lies closer to the plane stress handbook solution. Apart from small differences, all three methods thus show comparable results for this case.

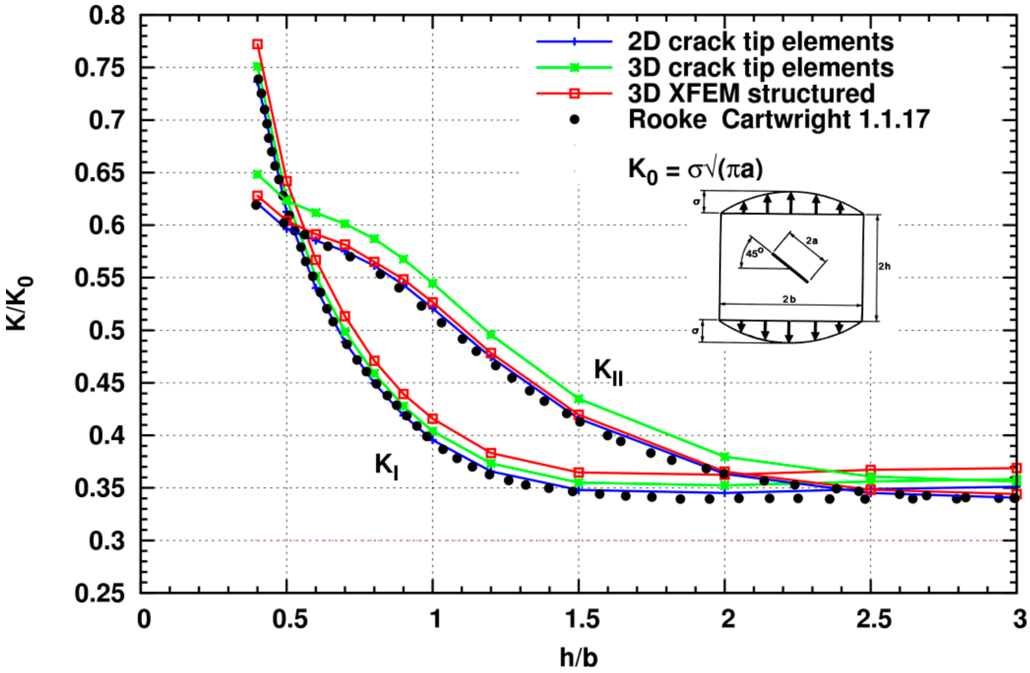

Figure 10: Computed normalised SIF for a central slant crack in a rectangular sheet loaded by a parabolic varying tensile stress.

4 F-16 wing lead crack growth analysis

An advanced crack growth analysis was performed for the lead crack configuration that occurred in the test in the lower skin starting at the rectangular cutout at butt-line 71, depicted in figure 6, and compared with the wing test results and the damage tolerance life analysis performed by LM for this location. The objective was to obtain a good estimate of the remaining life of the wing structure and to determine possible necessary actions. A second objective was to determine the life prediction capability of such an analysis and demonstrate the highly automated framework presented in the previous section.

Mode I and mode II Stress Intensity Factor solutions were computed for each of the 23 unit actuators applied in the F-16 Block 15 wing test using a coarse grid finite element model obtained from LM that is available at NLR. For this, the model was significantly refined in the region of the lead crack configuration around the rectangular cutout. The analysis resulted in accurate SIF solutions instead of the simple handbook solutions normally applied, taking into account the correct wing structure, the true test loads and the realistic crack configuration including interaction effects.

These 23 Mode I and II SIF solutions together with the load sequence applied in the wing test for all 23 actuators were used in a crack growth analysis, using a special in-house developed crack growth tool Cragro++ [11], to predict the damage tolerance life and inspection interval of the F-16 wing for this most critical location.

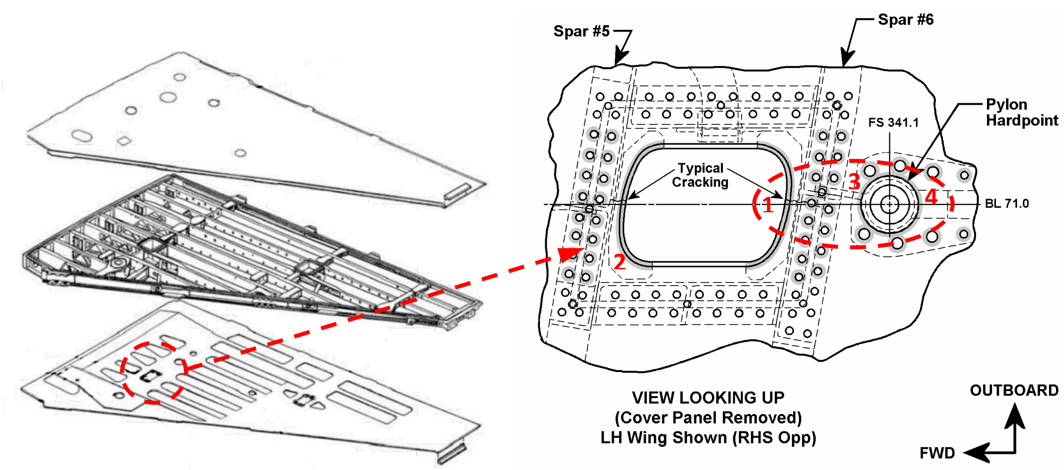

Figure 11: Schematised lead crack location, from DADTA LM.

The computation of the SIF solution for the lead crack of the F-16 Block 15 wing consists of a number of steps. The lead crack configuration that occurred in the test, depicted in figure 6, consisted of the lead crack in the lower skin starting at the rectangular cutout (location 1 in figure 11) at Butt-Line (BL) 71 growing to the spar number 6 and finally to the pylon hardpoint hole. During this period, other relevant cracks started at nearby locations in the rectangular cutout (location 2), pylon hardpoint hole (location 3 and 4) and in the underlying spar, depicted in figure 11, interacting with this crack. The spar crack runs under the fastener, one row away from the lead crack.

During the test non-destructive inspections (NDI) were performed at regular intervals both visually as well as with eddy current. Furthermore, marker load sequences were applied after every block of 500 flight hour (FH) leaving a characteristic striation pattern (marker bands) on the fracture surface that allows the crack growth to be tracked after the test by means of fractography. Figure 12 depicts the NDI (open symbols) and fractography data (closed symbols) that was obtained for the lead crack configuration. NDI data was only available after the crack was detected. Fractography data was only available when the marker bands were visible on the fracture surface.

Only one marker band was found for the spar crack together with the final crack length, from which the most likely crack growth line was constructed (black line), also supported by additional crack growth analyses. The crack sizes are plotted on a logarithmic scale resulting in more or less straight crack growth curves indicating exponential crack growth, apart from the NDO data close to breakthrough of the lead crack. Exponential crack growth has been frequently observed, by for instance DSTO and also NLR, especially for fighter aircraft spectra, see for instance [12].

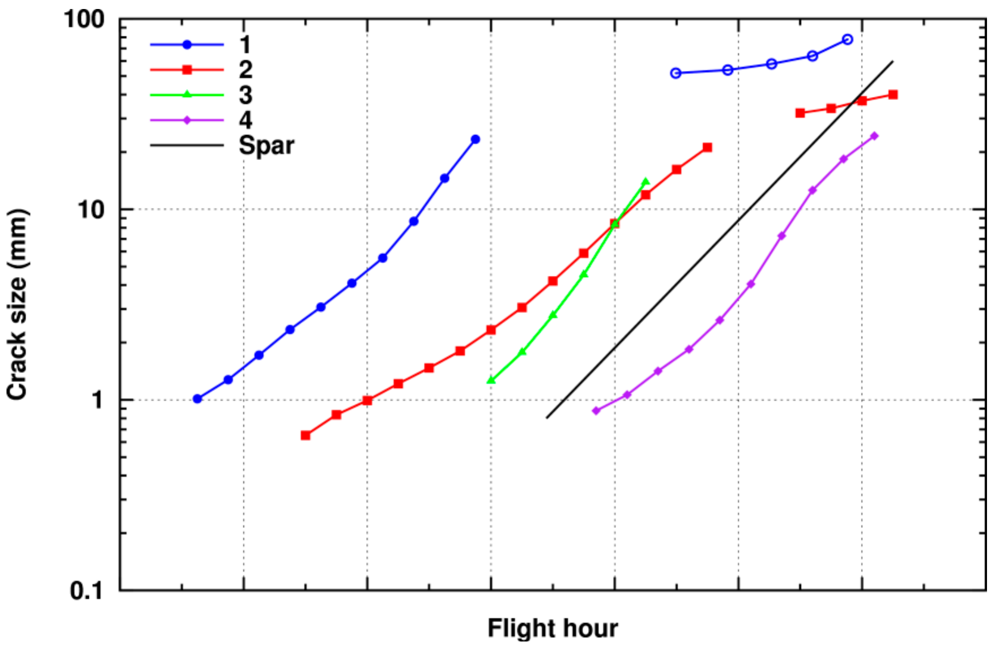

Figure 12: F-16 Block 15 wing test lead crack configuration NDI (open symbols) and fractography data (closed symbols).

As stated before, a small error in the SIF solution yields a much larger error in the computed life. Therefore the objective here was to compute an accurate SIF solution for the lead crack by means of finite element analyses. The left-hand-side wing part of the coarse grid finite element model of the F-16 Block 15 was used as a starting point. Besides the SIF computation, the wing model was applied in the definition of the wing interface in the test and the actuator loads. For each of the 23 actuators the displacement and stress/strain field for the wing in the test frame was computed for a unit load. Only part of the wing model was applied in the SIF computations, consisting of a region around the lead crack configuration. The corresponding sub-model is depicted in figure 13. The applied load on the edges is a displacement load since the sub-model is constraint by the surrounding wing structure. The edges of the sub-model were selected sufficiently away from the crack region, such that the displacement loads at the edges were not influenced by the crack development. For each of the 23 unit actuator loads the corresponding displacement boundary conditions on the sub-model edges were determined from the computed displacement fields for the whole wing model. For some actuators, partly acting in the area of the sub-model, additional forces were active as well. An example of the boundary conditions and internal forces is shown for actuator 1 in figure 13.

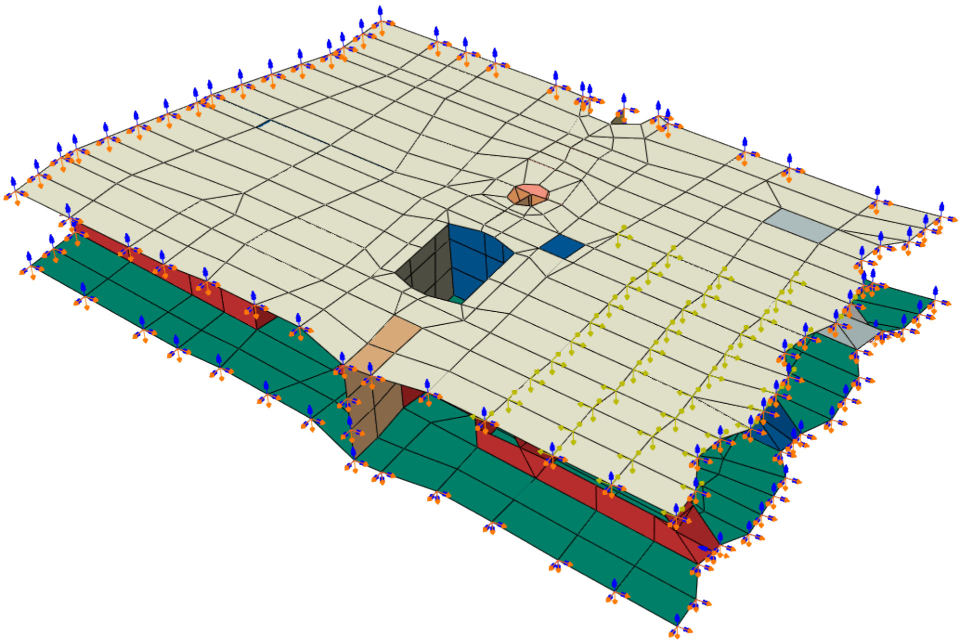

Figure 13: Example cgFEM sub-model of lead crack region including displacement and actuator loads.

The coarse grid finite element model sufficiently represents the stiffness of the wing structure, but cannot represent the local stress/strains field, which is required for an accurate SIF computation. Hence, the model was refined around the crack locations, for the skin and spar,

yielding an accurate model of the true geometry of the rectangular cutout together with a refined mesh. The thickness and offset values of the coarse grid model were hereby preserved. Apart from much more elements, 6900 instead of 37 elements for the skin and 2607 instead of 10 for the spar, the mesh is also build from quadratic instead of linear shell elements further improving the accuracy. The quadratic mesh is also required for the SIF computation using crack tip elements to accurately represent the stress field around the crack tip.
Figure 14 shows the combined refined and coarse mesh of the final complete sub-model. Special tie-constraints were applied to connect the refined meshes with the coarse grid model. For the skin this consisted of a tie-constrain to connect the outer edges, tie constraints to connect the skin to the underlying spars at the areas were the fasteners are located and tie-constraints to connect the skin and spars at the fasteners were cracks were located. These last constraints were updated once the crack tip arrives at the fastener hole. The tie-constraint for the corresponding fastener hole then was released to simulate a non-load transferring fastener.

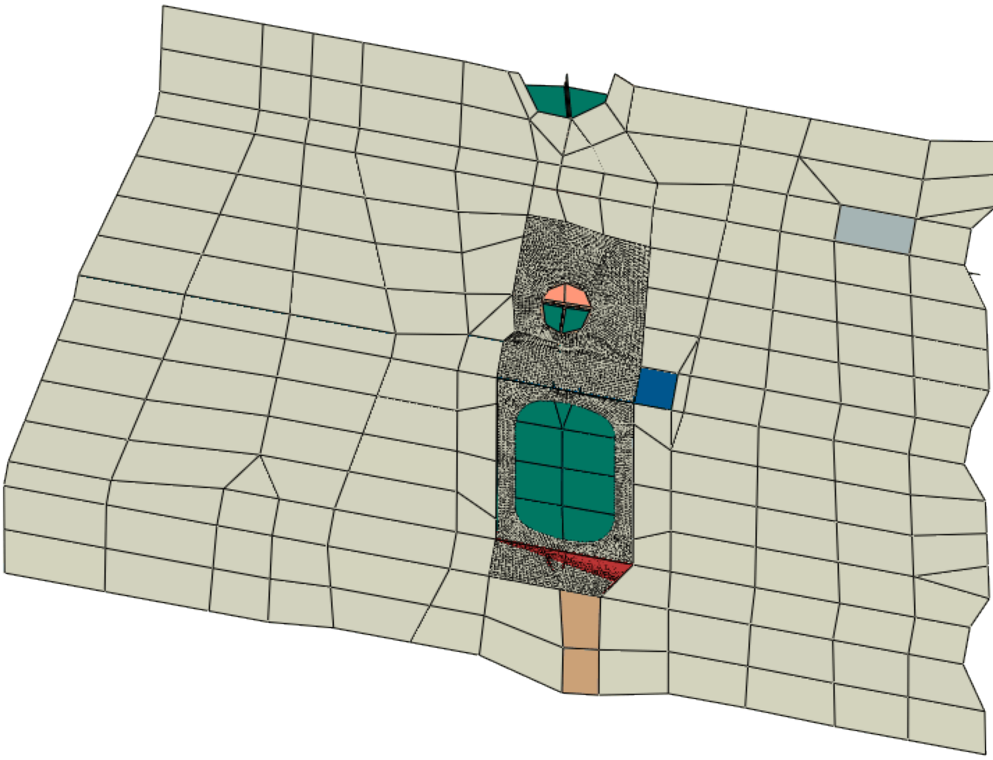

Figure 14: F-16 wing sub-model with refined mesh in the lead crack region.

The refined sub-model was applied in the SIF solution computation with the lifing framework. From the crack growth data obtained in the wing test, as depicted in figure 12, various crack configurations were determined for different stadia of the lead crack. The crack configurations hereby consisted of the lead crack size that grows from the rectangular cutout to the pylon hard

point and the corresponding crack sizes of the other three skin cracks and the crack in the spar. For this the exponential behaviour of the crack growth curves was applied allowing a realistic determination of the various crack sizes for a given lead crack size. The maximum crack advance was limited to 2 mm to get a detailed SIF solution. At start, the SIF solution for the lead crack is not influenced by the other cracks, however, for larger crack sizes it does. The crack paths for the different cracks consist of straight lines coinciding as much as possible with the crack paths observed in the wing test. Around each crack tip a focused-mesh was created that consist of a ring of crack-tip elements, surrounded by a number of additional rings of rectangular elements for accurate computation of the SIF solution.
Fastener holes were not modelled, because of the interference fit of the fasteners in the real structure causing a beneficial plastic deformation of the skin material around the fastener hole. The hole can therefore be seen as a filled hole. When the crack arrives at a fastener hole, the tie-constraint connecting the skin and underlying spar is released assuming no more load transfer is possible by the fastener. This is the case for the lead crack, the second crack in the left lower corner of the rectangular cutout and the crack in the spar. For the spar crack it is assumed that the fasteners keep on carrying load until the flange is fully cracked.
Because the fastener holes are not modelled, the crack is grown across the fastener hole, while in reality the crack will need time to reinitiate at the other side where it will restart as a corner crack. Hence, the computed SIF solution is somewhat conservative at these locations.

Since the finite element model was built from shell elements, all cracks are through cracks. However, the SIF solution is corrected for the fact that the lead crack starts as a corner crack by using the ratio of a known SIF solution for a corner and through crack. The crack becomes a through crack when the size equals the thickness of the skin.

Figure 15 depicts the computed normalised mode I and II SIF solution for the skin cracks for actuator 7, the colours refer to the crack locations as indicated by the insert. Each dot in the plot represents a separate finite element analysis, requiring: automatic crack advance, regeneration of the (focused) mesh, solution of the problem, computation of the J-integral and extracting the results. The overall behaviour shows a decreasing SIF solution due to loss of stiffness of the structure and load transfer to the spar. Crack interaction effects and release of fasteners when the crack arrives at a fastener hole causes a slight increase in the SIF solution.

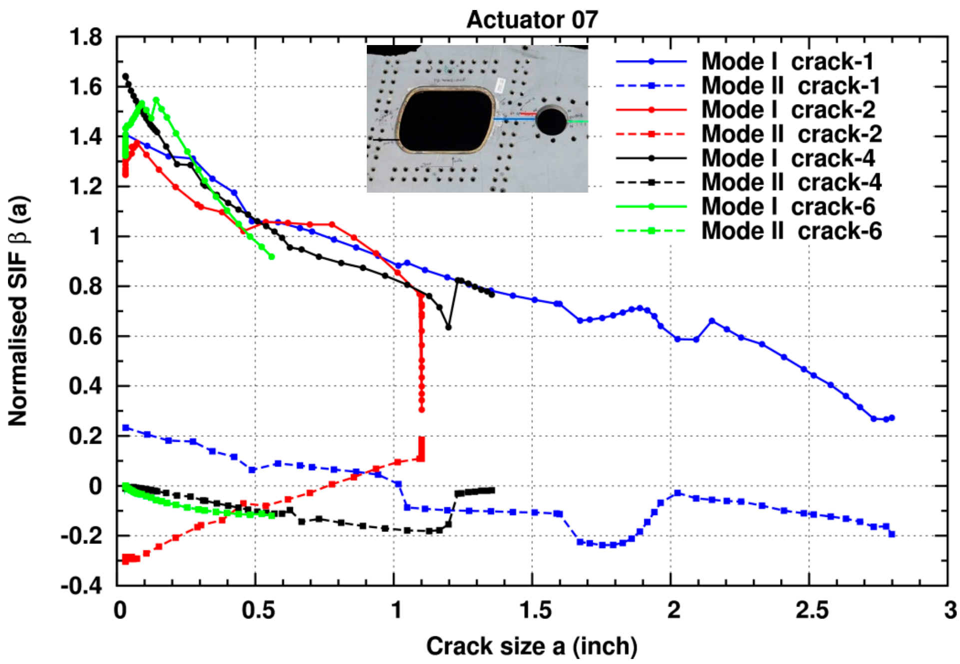

Figure 15: Normalised mode I and II SIF solution for skin lead crack configuration, actuator 7.

Figure 16 and 17 depict the mode I, respectively, mode II SIF solutions for all 23 actuators, showing a more or less similar behaviour. All SIF solutions are normalised with $K_0=\sigma\sqrt{\pi a}$ with a unit actuator load. The difference in SIF level is due to the location of the actuator. The upper curves correspond to the actuators around the tip of the wing (see also figure 2) and the bottom curves to the actuators near the wing root for which a unit load causes much less stress in the region of the rectangular cutout. Both mode I and mode II normalised SIF solutions are used in the crack growth life analysis discussed next.

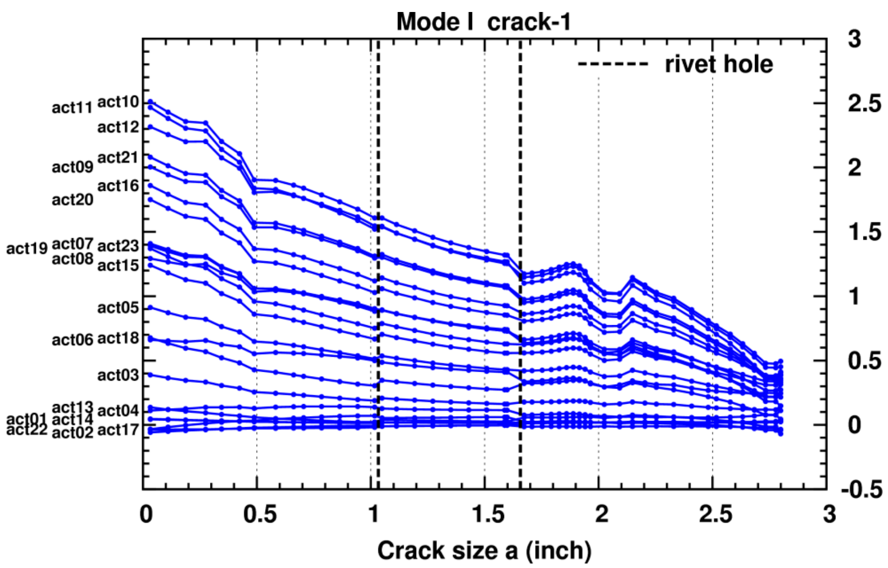

Figure 16: Normalised mode I SIF solution for skin lead crack for all actuators.

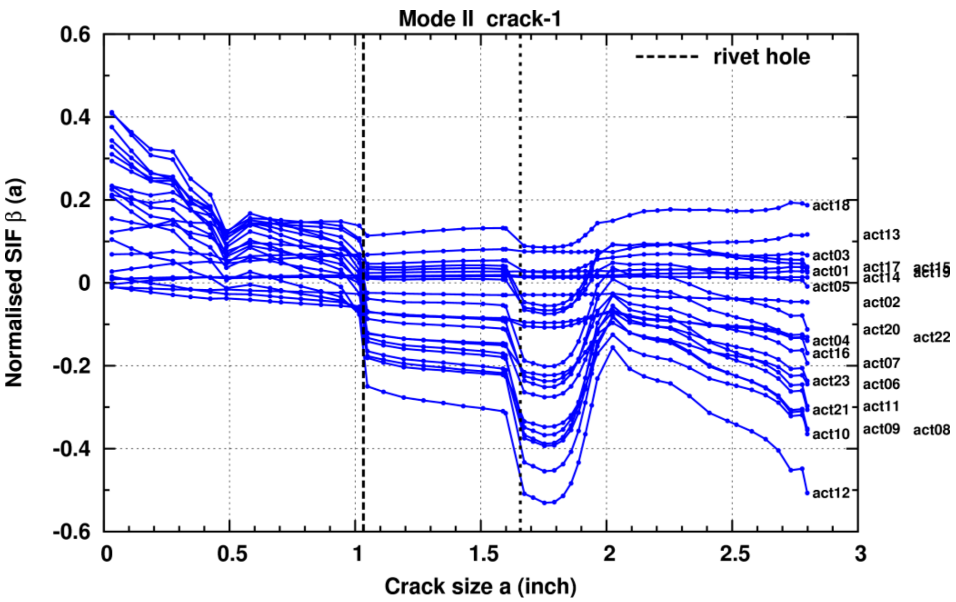

Figure 17: Normalised mode II SIF solution for skin lead crack for all actuators.

The RNLAF baseline load spectrum as applied by LM in the DADTA was used to derive the actuator loads of the wing test. The wing test results therefore can be directly compared with the LM DADTA result and also with NLR analysis result. The actuator spectra as applied in the test were used in the crack growth life analysis, however, these spectra contain intermediate points and intermediate load cycles (range pairs) which both needs to be removed. Hence, the spectrum was filtered to remove the intermediate points and rainflow counted to remove the intermediate cycles. Normal rainflow counting destroys the order of the cycles in the stress spectrum, which affects the history and therefore the retardation effects. Hence, an indexed rainflow filter scheme was applied that preserves the location of the peaks in the spectrum.

The same crack growth model as used by LM in their DADTA was applied, using the same material properties. The main difference with respect to the LM analysis is the applied SIF solutions, which are the finite element based solutions presented in the previous section instead of stress corrected handbook solutions as applied by LM. Also the stress spectrum is directly derived from the actuator load spectrum as applied in the test, which was similar as the stress spectrum applied by LM since both are derived from the same baseline load spectrum. This was checked afterwards.

The results of the crack growth life analysis are presented in figure 18 (in blue) together with the wing test results (in black) and LM crack growth results (in green). The red line denotes the critical crack size which is defined as functional impairment due to fuel leakage. The following observation can be made:

- From the wing test a damage tolerance life, starting with an initial crack size of 0.05 inch up to functional impairment, of 4100 FH is obtained.
- The LM crack growth result predicts the crack growth life up to the first fastener hole reasonably well, although predicting a somewhat faster crack growth. The crack growth from the first fastener to the second fastener is much slower than observed in the test causing a much longer (un-conservative) damage tolerance life of 10,500 FH up to functional impairment. The much slower crack growth cannot be explained by the neglection of the spar crack, having a limited effect on the crack growth up to the second fastener as can be observed from the NLR analysis results, but most likely is caused by the accuracy of the applied SIF solution and the neglection of the cracked spar.
- The NLR analyses predicts a damage tolerance crack growth life up to functional impairment that matches well with the corresponding life found in the test (without any tweaking), although the shape of the crack growth curve differs. The shape of both LM and NLR crack growth curves do not match the crack growth curve coming from the wing test. The reason for this is that the exponential crack growth as observed in the wing test cannot be represented well by conventional crack growth models. The remaining crack growth life after functional impairment, not of interest for the damage tolerance life prediction, consists of growth to the second rivet hole and up to the pylon hard point. The predicted life is small, whereas in the test a longer life was observed represented by the NDI data, although this data unfortunately could not be fully correlated with the fractography data and may possibly shift somewhat to the left. The most likely explanation for the larger test life is that a large part of the load is transferred to the remainder wing structure not taken in to account in the presented SIF analyses.

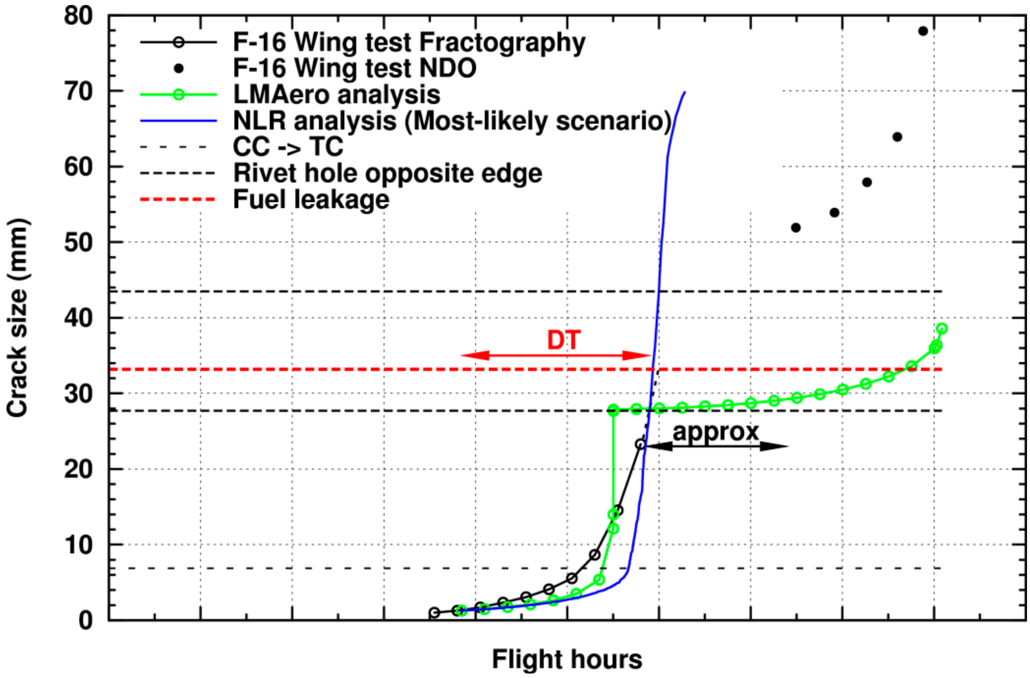

Figure 18: Crack growth life for the lower wing skin rectangular cutout lead crack.

5 Conclusions

In order to verify the current estimates for the service life and maintenance requirements of the RNLAF and FACh F-16 Block 15 wings, a durability test has been conducted at NLR on the left hand side wing of a decommissioned F-16 Block 15 aircraft of the RNLAF that had accumulated 4,200 operational flight hours. This durability test aimed to grow in-service cracks of sub-detectable size to a size where they could readily be detected. The main objective of the test was to determine if the ex-service wing contained damage not accounted for in the durability test programme of the 1970s by General Dynamics or in the analyses of Lockheed Martin. Other objectives were to generate data that can be used for an assessment of the current maintenance programme and to establish the most critical locations and the most likely fail scenario.
Upon completion of the damage enhancement test the wing had been subjected to a load spectrum that was equivalent to a total of 18,200 RNLAF flight hours (incl. the 4,200 service hours). During the test a number of fatigue cracks had initiated and grown, some of them to a significant size. The wing was able to sustain the three limit load cases that were applied after the durability test, however.

A lifing framework was developed at NLR to enhance the life prediction for complex geometries and/or complex loads by improving the accuracy of the applied stress intensity factor (SIF) solution. This can be done with the aid of the finite element method. The disadvantage is that the calculation of such a solution can be very time consuming. Due to the increase in computer capacity, the better integration of computer aided design and finite element tools, and new numerical algorithms, the computation of a SIF solutions today can be done much more efficiently. A framework was developed with which a crack growth life analysis for a complex geometry and load can be performed much more efficiently, automating the different steps as much as possible, especially the time consuming parts. The framework provides a coupling of an (integrated) CAD tool to define parameterised crack geometries, a finite element tool to compute the SIF solution and an advanced crack growth tool to compute the life of the structure.
The framework was verified by several one-dimensional and two-dimensional crack geometries. The SIF solution was automatically computed and compared against handbook solutions. The framework can also compute the SIF solution on-the-fly in a crack growth analysis, calling a finite element program from inside the crack growth program when a new solution is required, for instance enabling the SIF computation in case of a change in crack shape or crack bi-furcation such as in an integrally stiffened panel.

The framework was successfully applied to predict the damage tolerance life for the complex lead crack configuration that occurred in the full scale fatigue test on an F-16 Block 15 wing of the RNLAF.

Acknowledgement

The research presented was funded by a shared effort of the Royal Netherlands Air Force and the Chilean Air Force.

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